

N72-26966

NASA TECHNICAL
MEMORANDUM

NASA TM X-2564



NASA TM X-2564

CASE FILE
COPY

COMPARISON OF HEAT-TRANSFER
CHARACTERISTICS OF TWO
AIR-COOLED TURBINE BLADES
TESTED IN A TURBOJET ENGINE

by *Gerald E. Fallon and John N. B. Livingood*

Lewis Research Center

Cleveland, Ohio 44135

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JUNE 1972

1. Report No. NASA TM X-2564	2. Government Accession No.	3. Recipient's Catalog No.	
4. Title and Subtitle COMPARISON OF HEAT-TRANSFER CHARACTERISTICS OF TWO AIR-COOLED TURBINE BLADES TESTED IN A TURBOJET ENGINE		5. Report Date June 1972	
7. Author(s) Gerald E. Fallon and John N. B. Livingood		6. Performing Organization Code	
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		8. Performing Organization Report No. E-6822	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		10. Work Unit No. 764-74	
15. Supplementary Notes		11. Contract or Grant No.	
		13. Type of Report and Period Covered Technical Memorandum	
		14. Sponsoring Agency Code	
16. Abstract <p>Two air-cooled blades were tested to gas temperatures of 1682 K (2568° F) with coolant temperatures to 680 K (764° F). Both average and local blade temperatures were correlated over the range of operating conditions. Calculated and measured blade wall temperatures are compared. Potential operating turbine-inlet temperatures of each blade are discussed.</p>			
17. Key Words (Suggested by Author(s)) Turbine cooling; Heat transfer; Spanwise fins; Impingement cooling; Convection cooling; Cooled airfoil temperature correlations		18. Distribution Statement Unclassified - unlimited	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 28	22. Price* \$3.00

* For sale by the National Technical Information Service, Springfield, Virginia 22151

COMPARISON OF HEAT-TRANSFER CHARACTERISTICS OF TWO AIR-COOLED TURBINE BLADES TESTED IN A TURBOJET ENGINE

by Gerald E. Fallon and John N. B. Livingood

Lewis Research Center

SUMMARY

Heat-transfer characteristics of two internal cooling configurations for air-cooled turbine blades were investigated experimentally. These tests were made in a modified J75 turbojet engine capable of operating at average turbine-inlet temperatures to approximately 1644 K (2500° F). Both average and local midspan blade temperatures were correlated over a wide range of operating conditions. One of the blades tested was a simple cast blade with a radial flow-through configuration. The other was a spanwise-passage blade containing spanwise convection cooling passages, an impingement-cooled leading edge, and a slotted convection-cooled trailing edge.

For the range of engine operating conditions investigated, it was found that the spanwise-passage blade cooled considerably better than the simple cast blade. The cooling effectiveness (defined as the effective gas temperature minus the average midspan blade temperature divided by the effective gas temperature minus the coolant temperature at the inlet to the blade) of the spanwise-passage blade was about 20 to 35 percent higher than that of the simple cast blade. This better cooling of the spanwise-passage blade was expected because this blade incorporated a more sophisticated cooling design which included impingement cooling of the leading edge and more coolant-side heat-transfer surface area in the midchord region. At engine design conditions the spanwise-passage blade was predicted to have a chordwise gradient of about 167 K (300° F), about the limit considered allowable.

The potential operating turbine-inlet temperatures of the two blades were calculated empirically for an assumed maximum blade wall temperature of 1254 K (1800° F). At a ratio of coolant to gas flow of 0.03 and a coolant temperature of 922 K (1200° F), it was found that the spanwise-passage blade could operate at a turbine-inlet temperature of 1600 K (2423° F) and the simple cast blade at 1520 K (2276° F). At a potential operating turbine-inlet temperature of 1644 K (2500° F), it was found that the spanwise-passage blade could operate with half as much coolant flow as the simple cast blade, which indicates the reduction in coolant flow possible by utilizing a more refined and complicated design.

INTRODUCTION

The heat-transfer characteristics of two internal cooling configurations for air-cooled turbine blades were investigated experimentally. This investigation is part of a series of vane and blade cooling investigations being made at the NASA Lewis Research Center to determine the cooling effectiveness of advanced air-cooled vanes and blades.

Significant increases in turbine engine output can be achieved by operating at elevated turbine-inlet temperatures. With present day turbine materials this high-temperature operation can be achieved only through cooling the turbine components. Turbine rotor blades are the most critical components of the turbine assembly because they are subject to both high gas temperatures and high centrifugal loads. Turbine cooling is required for average turbine-inlet temperatures in excess of about 1282 K (1850° F). To increase this operating temperature appreciably, efficient cooling concepts must be designed into the turbine rotor blades as well as the stator vanes. In order to evaluate various internal cooling concepts adequately it is desirable to develop techniques for the prediction of blade metal temperatures. Ideally, wall temperatures should be determined analytically. Empirical correlations such as those developed in reference 1 have been used successfully in predicting blade wall temperatures. Experimental comparison of internal cooling concepts has also been used to evaluate various methods of turbine cooling.

This present investigation provides such an experimental comparison for two widely differing cooling configurations. One configuration, hereinafter called the simple cast blade, is a rugged, simple, one-piece casting with radial coolant passages that provide for a single pass of the cooling air with tip discharge. There is no more machining required on this configuration than if it were a solid, uncooled turbine blade. This configuration is used as a slave blade on the test engine. The second configuration tested is considerably more complicated. It was designed to minimize the cooling air required for a purely convection cooled blade. Little consideration was given to providing for simplicity in manufacture. This configuration, hereinafter called the spanwise-passage blade, utilized radial (spanwise) fins in the midchord region and an impingement-cooled leading edge. The blade tip was capped, and all the cooling air was forced to exit through and cool a slotted trailing edge. The purpose of comparing these widely differing configurations was to find the amount that the coolant flow could be reduced (compared to the simple cast blade) for a given gas temperature or the amount that the gas temperature could be increased for a given coolant flow by utilizing effective cooling schemes with minimum regard to fabrication complication. The blades investigated had a span of about 10.16 centimeters (4 in.) and a chord of about 4.06 centimeters (1.6 in.).

The experimental investigation was conducted in a modified J75 turbojet engine capable of operating at average turbine-inlet temperatures to somewhat in excess of

1644 K (2500° F). Blade temperatures were obtained for ratios of cooling air to combustion gas flow ranging from 0.019 to 0.155 at turbine-inlet temperatures ranging from 1292 to 1682 K (1868° to 2568° F). At these temperatures the gas pressures at the turbine inlet ranged from a minimum of 20.67 to a maximum of 28.96 newtons per square centimeter (30 to 42 psia), and the turbine-inlet gas-flow rate varied from about 22.7 to 36.3 kilograms per second (50 to 80 lb/sec). Cooling-air temperatures ranged from room temperature to 680 K (764° F).

APPARATUS

Research Engine

The investigations described in this report were conducted in a modified J75 turbojet engine. Only the high-pressure spool of the basic engine was used.

A detailed description of the research engine, the test facility, and the data acquisition system, is given in reference 2. A schematic of the turbine section of the research engine is shown in figure 1. The major modifications applied to the J75 high-pressure spool to produce the research engine were the following:

- (1) Replacement of the standard uncooled single-stage turbine with an air-cooled turbine
- (2) Provision for two separate and distinct cooling-air systems for both the stator and the rotor
- (3) Modification of the can annular combustor to operate continuously at a maximum average turbine-inlet temperature of 1644 K (2500° F)
- (4) Partial reblading of the standard compressor to match the air-cooled turbine better

The turbine blade row contains 76 air-cooled blades. Five adjacent blades are test blades; the remaining 71 are "workhorse" or slave blades. These five test blades were tested in conjunction with five test vanes in the vane row as described in reference 2.

The design operating conditions of the research engine were an average turbine-inlet temperature of 1644 K (2500° F) and a turbine-inlet pressure of 31.0 newtons per square centimeter (45 psi). The external cooling-air system associated with the research engine was capable of supplying air to the vanes and blades at a maximum temperature of 921 K (1200° F). Both turbine-inlet and cooling-air inlet temperature values are representative of advanced high-temperature engine designs. The values of the other engine design operating conditions are basically inherent in the modified J75 engine.

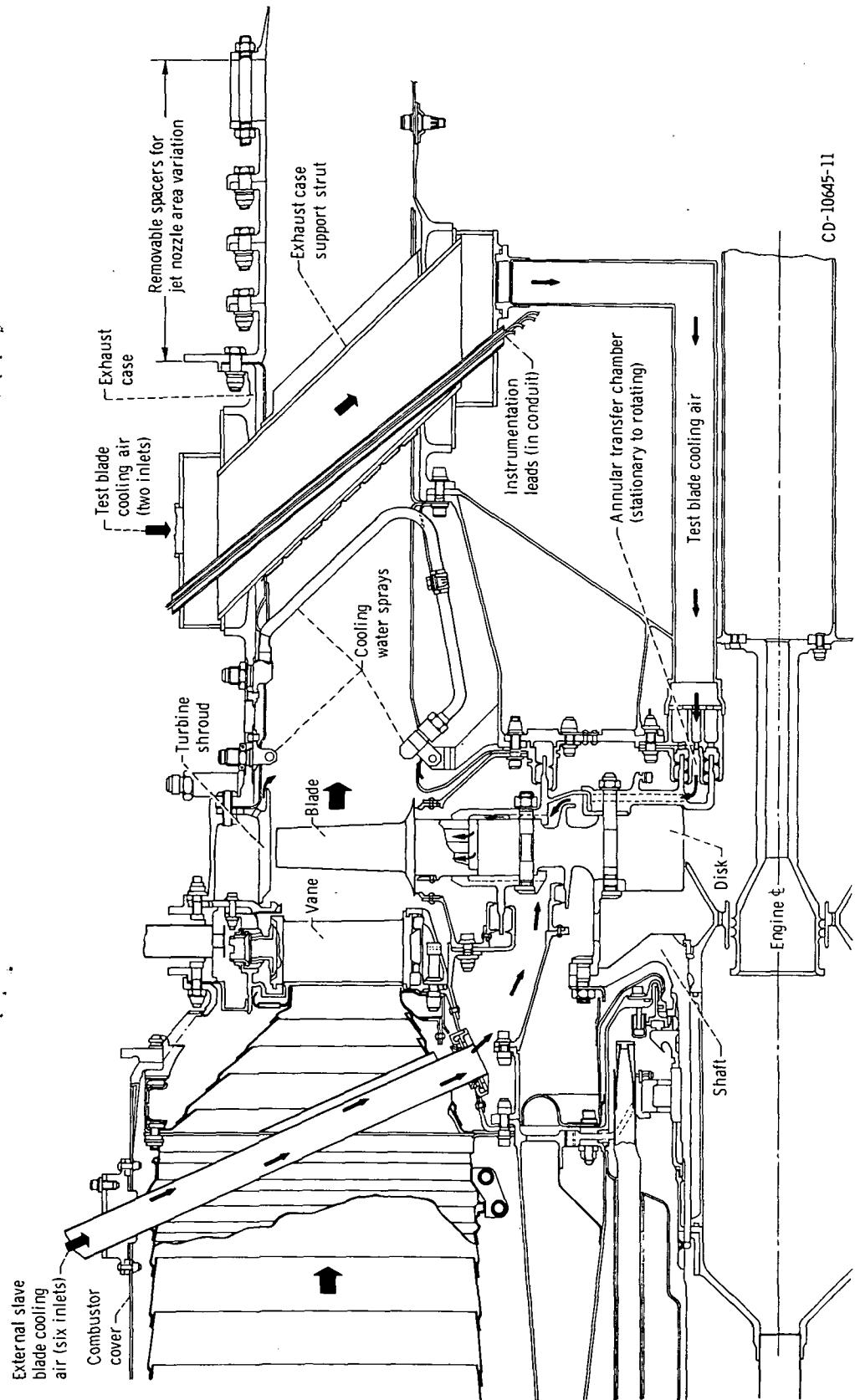


Figure 1. - Schematic drawing of hot section of turbine cooling research engine.

Test Blades

Two blade internal cooling configurations were tested. The external profiles and the bases of both test configurations were identical. The span of the airfoil portion was about 10.16 centimeters (4.0 in.), and the midspan chord was about 4.06 centimeters (1.6 in.).

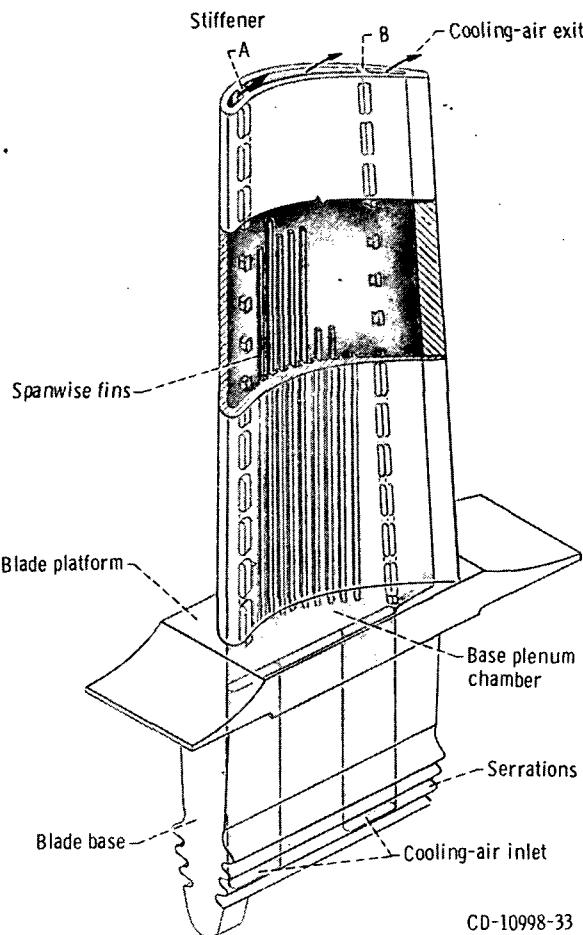
Simple cast blade. - A cutaway view of the blade referred to in this report as the simple cast blade is shown in figure 2(a). Cooling air entered the blade through two cooling passages in the base and flowed radially outward and exited through the blade tip. Stiffeners A and B divided the cooling cavity into three regions. Each stiffener contained 13 slots which permitted cooling air to pass between the leading edge and midchord regions and between the trailing edge and midchord regions of the blade. Each slot was 0.63 centimeter (0.25 in.) long; the slots were 0.10 and 0.076 centimeter (0.04 and 0.03 in.) wide in stiffeners A and B, respectively. Stiffeners A and B had about 80 percent open flow area. The figure also shows 19 spanwise fins of varying lengths in the midchord region of the blade. Nine of these fins were on the blade suction side, and 10 on the blade pressure side of the central cooling passage.

The simple cast blades were cast in a single piece and were supplied by the contractor from whom the test engine was purchased.

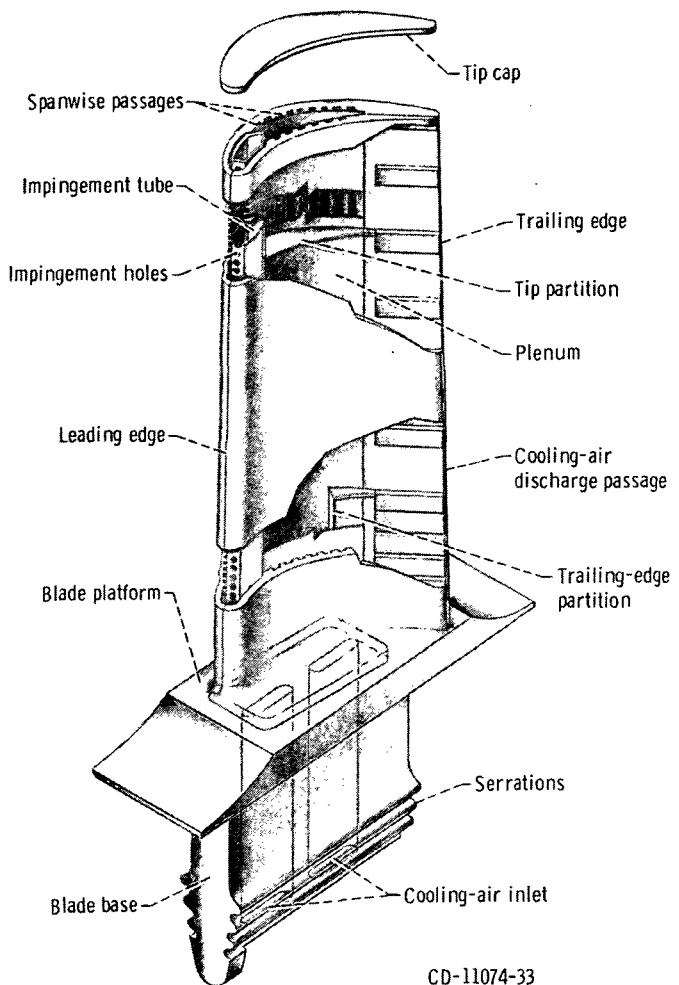
Spanwise-passage blade. - A cutaway view of this blade is shown in figure 2(b). Cooling air flowed through two passages in the blade base to a platform plenum chamber. From this chamber the cooling air flowed into distinct radial cooling passages to cool various chordwise regions of the blade. In the leading-edge region the cooling air entered an impingement tube; from this tube, the air flowed through 80 impingement cooling holes of 0.076-centimeter (0.030-in.) diameter along the blade span. The impingement air, after impinging on the internal surface of the blade leading edge, flowed around the impingement tube into a central plenum chamber and then was discharged through the slotted trailing-edge passage.

In the blade midchord region, cooling air flowed radially outward toward the blade tip through 15 spanwise passages, 9 on the blade suction surface and 6 on the blade pressure surface. These passages varied in width from 0.10 to 0.127 centimeter (0.04 to 0.05 in.) and were tapered in depth from 0.076 centimeter (0.03 in.) at the blade base to 0.063 centimeter (0.025 in.) at the blade tip. The cooling air flowing through the spanwise passages was turned through a 180° angle at the tip and flowed into the central plenum. This cooling air was discharged through the upper 2.54 centimeters (1 in.) of the slotted trailing edge because of the partition located 2.54 centimeters (1 in.) from the blade tip.

To ensure adequate cooling in the blade trailing-edge base region, another partition was inserted into the blade to partition off the lower 2.54 centimeter (1 in.) of the blade



(a) Simple cast blade.



(b) Spanwise-passage blade.

Figure 2. - Cutaway views of blades.

trailing edge. This region was cooled directly by cooling air entering from the base plenum and flowing out through this split trailing edge. The split trailing edge has two partitions in the upper 2.54 centimeters (1 in.), six partitions in the lower 2.54 centimeters (1 in.), and four partitions in the central part.

These blades were designed and fabricated at the Lewis Research Center and were intended to be purely a research tool. To make this blade, two solid blades cast of Udimet 700 with the proper airfoil shape were used. The suction side of one blade and the pressure side of the other blade were electrical-discharge machined to the desired internal cooling configurations. Thin sheet metal walls were brazed to radial fins of each airfoil half, and the impingement tube was also brazed to one of the airfoil halves.

The pressure and suction side portions of the blade were then electron-beam welded at the leading- and trailing-edge regions to form a single assembled blade. After welding, the final machining of the blade platform, base, and serrations was accomplished.

The design conditions of this blade were a turbine-inlet temperature of 1539 K (2300° F), a 922 K (1200° F) coolant temperature, a coolant- to gas-flow ratio of 0.03, and a turbine-inlet pressure of 31 newtons per square centimeter (45 psia). The allowable average metal temperature for the Udimet 700 blade was 1089 K (1500° F).

INSTRUMENTATION

The normal complement of operational instrumentation was used on the research engine. In addition, the following research instrumentation was used to measure conditions of specific interest.

Gas and Coolant Instrumentation

Eight actuating thermocouple probes were located at various circumferential positions ahead of the stator vanes to measure the turbine-inlet temperature. Each of these probes was actuated to traverse radially across the gas stream to provide temperatures at nine similar spanwise (or radial) locations to determine a representative average turbine-inlet temperature.

It was originally intended that the coolant-flow rate to the test blade proper would be determined by using a Venturi-measured flow rate in the supply line external to the engine and correcting it for leakage at the labyrinth seal system and at the blade base serrations. During testing it was found that uncertainties in coolant flow rates due to leakages under rotating conditions resulted in unreliable blade coolant-flow rates. Consequently, it was necessary to determine the test blade coolant-flow rate, as explained in the section ANALYSIS METHODS.

The temperature and static pressure of the test blade cooling air were measured in the test blade cooling-air annular chamber immediately before the coolant entered the rotating turbine disk (fig. 1). The cooling-air temperature was also measured at the base of the rotor blades just prior to entering the blade cooling passages.

Blade Instrumentation

Because of structural considerations and physical limitations, no more than five

thermocouples were installed on any one blade. In order to obtain a satisfactory number of experimental wall temperatures, it was therefore necessary to instrument all five of the test blades. For each configuration tested, most of the thermocouples were located at or near the midspan section of the blade.

The spanwise passage blade, because of its relatively thin wall, could not be instrumented meaningfully at any section past the 3.81-centimeter (1.5-in.) section. At this position, the blades were instrumented with 19 thermocouples. This section will be referred to as the midspan on the NASA designed blade.

The simple cast blade was instrumented at the hub, mean, and tip. Eight of the 14 thermocouples installed on the blades were at the midsection, that is, the 5.08-centimeter (2.0-in.) section.

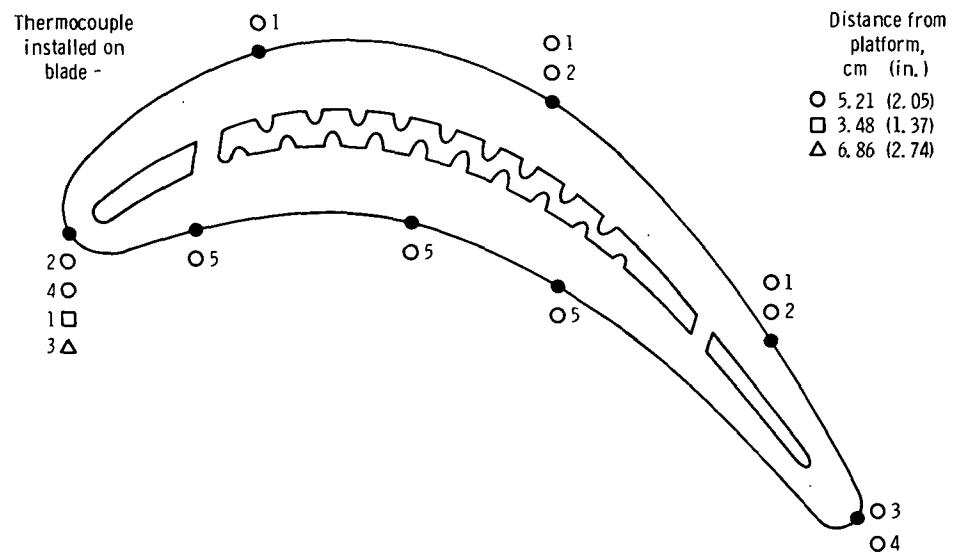
Sheathed thermocouples, consisting of Chromel-Alumel thermoelements and magnesium oxide insulation in a 0.051-centimeter- (0.020-in.-) diameter Inconel 600 tube, were installed in radial grooves machined in the blade surface. All sheathed thermocouples used for these tests were formed with a closed and grounded hot junction. The forming and installing techniques are discussed in reference 3.

Simple cast blade. - Figure 3(a) shows a composite instrumented simple cast blade. A total of 14 thermocouples formed this blade composite; 8 at different chordwise thermocouple positions were used at the midspan section (four midspan thermocouples were duplicated on different blades of the five-blade test pack). One thermocouple was located near the blade base and one near the blade tip. The numbers associated with the thermocouples identify the particular blade on which the thermocouple was installed, numbering from left to right in the five-blade test pack as viewed from the front.

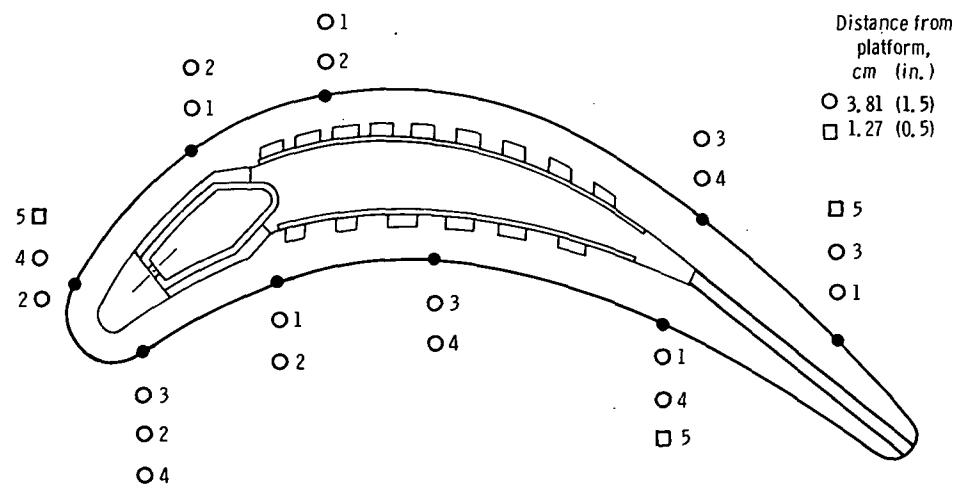
Table I gives the exact location of each blade midspan thermocouple.

Spanwise-passage blade. - A total of 23 thermocouples were installed on this five-blade test pack (see fig. 3(b)). Four of these thermocouples were located near the blade base, and the remainder near the blade midspan section. Because the trailing edge pressure surface wall was so thin, no thermocouples could be installed in it. One was installed on the suction-surface trailing edge. Thermocouples were installed at a total of nine chordwise positions at the midspan section; each position had thermocouples located on at least two of the blades. As for the simple cast blade, the numbers in figure 3(b) identify the blade on which the thermocouple was installed. Table I gives the location of each blade midspan thermocouple.

It should be noted that no thermocouples were placed at the leading edge of the spanwise-passage blade. This experimental blade was made in two halves, and one of the seams which joined these halves was located at the leading edge. For structural reasons, it was believed desirable to avoid grooving and installing a thermocouple in the welded region.



(a) Simple cast blade.



(b) Spanwise-passage blade.

Figure 3. - Composites of thermocouple locations.

TABLE I. - MIDSPAN BLADE THERMOCOUPLE LOCATIONS

Blade side	Simple cast blade ^a			Spanwise-passage blade ^b		
	Distance along surface from stagnation point, x		Ratio of distance along surface to surface length, x/L	Distance along surface from stagnation point, x		Ratio of distance along surface to surface length, x/L
	cm	in.		cm	in.	
Suction	0.076	0.030	0.015	0.128	0.150	0.076
	1.560	.613	.301	1.382	.544	.276
	2.715	1.068	.527	2.140	.842	.428
	4.14	1.630	.800	3.754	1.478	.751
	5.16	2.030	1.00	4.370	1.72	.875
Pressure	0.394	0.155	0.092	0.300	0.188	0.0656
	1.437	.565	.330	.880	.346	.195
	2.420	.951	.558	1.402	.552	.311
	-----	-----	-----	2.780	1.095	.617

^aSuction-side length, 5.08 cm (2.037 in.); pressure-side length, 4.33 cm (1.704 in.).

^bSuction-side length, 4.991 cm (1.965 in.); pressure-side length, 4.511 cm (1.776 in.).

EXPERIMENTAL PROCEDURE

Five instrumented test blades of one of the cooling configurations were installed in the test blade slots on the turbine rotor. The engine was operated over a range of turbine-inlet temperatures from 1292 to 1682 K (1868° to 2568° F) with the engine speed varying from about 6727 to 8800 rpm. Three blade cooling-air-inlet temperatures were investigated at these engine conditions, with most of the testing being conducted with cooling air at a temperature of about 380 K (224° F). Individual series of test points were also obtained for cooling-air inlet temperatures of about 500 and 700 K (440° and 800° F). For a given series, a change of coolant-flow rate was the primary variable from one test point to the next. During engine operation, reasonable amounts of air were supplied to the slave components in order to cool them sufficiently and promote extended component life since these slave components were to be used in future turbine cooling research.

Since most of the pertinent instrumentation was located at or near the midspan position, the turbine-inlet temperature was determined as the average of the measurements of the eight actuating thermocouple probes at the midspan position. A summary of the average engine operating conditions for each series of test points is given in tables II and III for the two test blade configurations. The symbols used in the figures for the various test series are shown in tables II and III.

TABLE II. - SUMMARY OF ENGINE OPERATING CONDITIONS FOR SIMPLE CAST BLADE

Test series	Engine speed, rpm	Average turbine-inlet temperature, $T_{t,i}$		Average turbine relative temperature, $T_{r,i}$		Average turbine-inlet pressure		Average cooling-air inlet temperature, T_c		Blade coolant- to gas-flow ratio, \dot{w}_c/\dot{w}_g	Symbols used in figures		
						N/cm ²	psia						
		K	°F	K	°F			K	°F				
1	7154 to 7185	1350	1971	1183	1670	21.35	31	347	165	0.023 to 0.155	□		
2	7997 to 8220	1515	2267	1315	1907	26.18	38	358	186	0.026 to 0.134	▽		
3	8068 to 8123	1508	2256	1331	1937	25.49	37	494	429	0.041 to 0.116	□		
4	8652 to 8695	1682	2568	1511	2261	28.24	41	380	224	0.038 to 0.108	△		

TABLE III. - SUMMARY OF ENGINE OPERATING CONDITIONS FOR SPANWISE-PASSAGE BLADE

Test series	Engine speed, rpm	Average turbine-inlet temperature, $T_{t,i}$		Average turbine relative temperature, $T_{r,i}$		Average turbine-inlet pressure		Average cooling-air inlet temperature, T_c		Blade coolant- to gas-flow ratio, \dot{w}_c/\dot{w}_g	Symbols used in figures		
						N/cm ²	psia						
		K	°F	K	°F			K	°F				
1	7626 to 7917	1292	1866	1183	1670	28.93	42	326	127	0.0194 to 0.091	△		
2	6727 to 7826	1382	2027	1242	1776	20.67	30	348	167	0.014 to 0.141	□		
3	8383 to 8765	1658	2524	1467	2180	27.56	40	397	254	0.026 to 0.111	◇		
4	8220 to 8492	1527	2288	1346	1962	25.83	37.5	494	429	0.028 to 0.081	○		
5	7820 to 7943	1368	2004	1214	1728	23.42	34	680	764	0.024 to 0.065	□		

The general engine operating procedure for a given series of test points was to start with a relatively large coolant-flow rate supplied to the test blades. For each successive test point in a series, the coolant-flow rate to the test blades was reduced. This procedure of reducing the coolant flow in a stepwise manner was continued until a maximum blade metal temperature of about 1200 K (1700° F) had been reached. For the stress-rupture characteristics of the blade material, the experimental temperature gradients, and the fabricating procedures used in the test blades, the 1200 K (1700° F) temperature level was considered the maximum safe operating temperature for the test blade configurations. After the minimum coolant-flow condition was reached, two additional test points were taken at intermediate coolant-flow rates to serve as check points for data taken previously. Usually about 10 test points were taken in each series. The experimental data were recorded on the laboratory digital data acquisition equipment.

ANALYSIS METHODS

Several methods for correlating experimental heat-transfer data were developed in reference 1. They were applied to data obtained from heat-transfer tests of a vane in references 1 and 4 and a blade in reference 5. The results presented in these references showed that the temperature difference relation

$$\varphi = \frac{T_{g,e} - T_w}{T_{g,e} - T_c} = f\left(\frac{\dot{w}_c}{\dot{w}_g}\right)$$

correlated the data as well as or better than the other correlations. (Symbols are defined in the appendix.) This correlation is also the easiest to apply of those developed in reference 1. Use of the correlation equation will be limited in this report to correlating average and local midspan blade wall temperatures. This limitation was made because a number of thermocouples sufficient to yield meaningful results were installed only at the midspan region of each blade tested.

The correlation was applied to the data for the two blade configurations tested. Local values of the measured midspan blade wall temperature T_w were used for local correlations. If more than one reading was obtained at any chordwise position, the average was used for this location. The integrated chordwise average measured midspan blade wall temperatures \bar{T}_w was used in place of T_w for the correlation of average data. For average correlations, the correlation parameter φ is redefined as $\bar{\varphi}$. The cooling-air temperature measured at the inlet to the blade base was used for T_c and the midspan turbine-inlet relative temperature $T_{r,i}$ was used as the effective gas temperature $T_{g,e}$ at the midspan section. This relative temperature was obtained by correcting the measured average midspan turbine-inlet temperature $T_{t,i}$ for the effects of stator vane cooling-air dilution and turbine rotation. The coolant flow and gas flow per blade were used for evaluating \dot{w}_c/\dot{w}_g .

As explained previously in the section INSTRUMENTATION, it was originally intended that the coolant-flow rate to the test blade would be determined by using a Venturi. However, because of the unreliable blade coolant-flow rates obtained under rotating conditions, a semiempirical method described in reference 5 was used to determine the test blade coolant flow.

The procedure discussed in reference 6 was used to extrapolate experimental blade temperatures from values obtained at low-gas-temperature conditions to corresponding blade temperatures at high-gas-temperature conditions.

RESULTS AND DISCUSSION

Spanwise Gas-Temperature Profile

A turbine-inlet gas temperature was measured at nine radial positions with each of eight circumferentially spaced turbine-inlet probes located in a plane in front of the vane row. Corresponding positions of each of the eight probes were averaged to obtain a circumferentially averaged turbine-inlet gas-temperature profile. A typical profile is shown in figure 4. Figure 4 shows that in the midspan region the three central positions

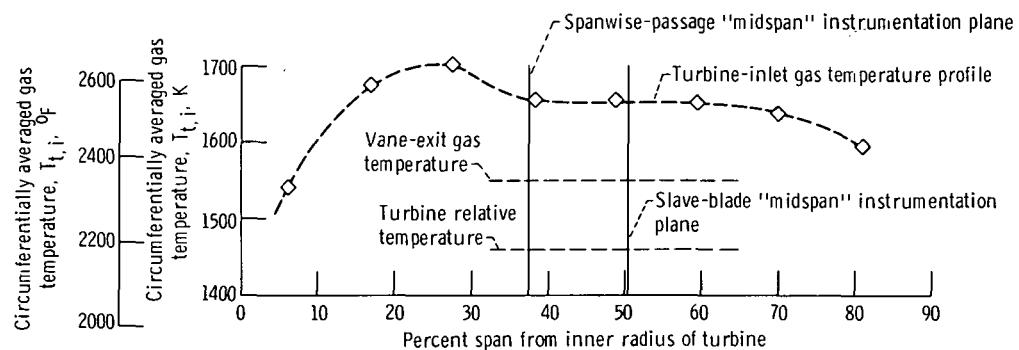


Figure 4. - Typical radial gas temperature profile at turbine inlet for research engine. Conditions given in table III (series 3).

are essentially of equal temperature. These temperatures were averaged over the eight probes to obtain an average midspan turbine-inlet gas temperature $T_{t,i}$. Because the analysis in this report is at the midspan only, this temperature is referred to as the turbine-inlet temperature hereinafter.

Also shown in figure 4 are two calculated temperatures for the particular series of test points, that is, series 3 of the spanwise finned-blade test data. The vane-exit gas temperature, which is lower than the turbine-inlet temperature primarily because of the dilution of the vane cooling air, is the higher of the two. The turbine-inlet relative temperature $T_{r,i}$ is the gas temperature relative to the rotating turbine. These temperatures were calculated from the average midspan turbine-inlet gas temperature and are only applicable over the midspan range indicated and for a specific vane cooling-airflow rate and turbine speed.

Two vertical lines in figure 4 indicate the "midspan" sections of each blade that were analyzed with respect to the spanwise gas-temperature profile, as discussed in the section INSTRUMENTATION. The line at 37.5 percent span represents the section analyzed on the spanwise passage blade, and the line at 50 percent span represents the section analyzed on the slave blade. Because the average midspan turbine-inlet gas tem-

perature is applicable in this spanwise region, it was justifiable to use this temperature to determine the vane-exit gas temperature and the turbine-inlet relative temperature used in the analysis of the midspan section of the blades.

The turbine relative temperature as a function of turbine-inlet temperature is shown in figure 5 for each series of test points listed in tables II and III. The solid line in this figure is a 45° line. The dashed line through the data has a slope of 0.90 on an absolute scale, which indicates that the turbine relative temperature is 90 percent of the turbine-inlet temperature for this particular test engine over the indicated range of turbine-inlet temperatures.

This relation between the turbine-inlet temperature and the turbine relative temperature is a function of the stator-exit Mach number, the rotor-inlet relative Mach number, and the dilution effects. For a design midspan stator-exit Mach number of 0.854, a design rotor-inlet relative Mach number of 0.475, and the range of slave-vane coolant dilution of the tests, the ratio of the turbine relative temperature to the turbine-inlet temperature is about 0.9.

The scatter in the data in figure 5 is due partially to the variation in the cooling of the slave vanes in order to promote extended slave-vane life, since these vanes are to be used with other groups of research vanes.

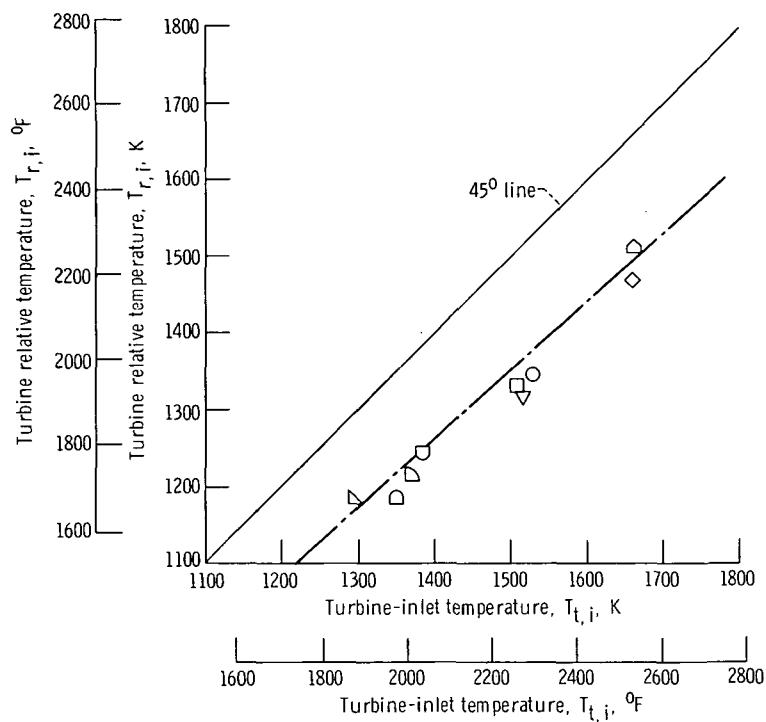


Figure 5. - Turbine relative temperature as function of turbine-inlet temperature for research engine. Conditions given in tables II and III.

Experimental Chordwise Blade Wall Temperature Distributions

Figure 6(a) shows the experimental midspan chordwise temperature distribution for each blade for approximately the same gas and coolant conditions. The simple cast blade data are from series 4 (see table II), and the spanwise-passage blade data are from series 3 (see table III). The average of the turbine-inlet temperatures for these series was 1670 K (2546° F), the average of the turbine relative temperatures was 1489 K (2221° F), and the average of the coolant temperatures was 388 K (239° F). The coolant- to gas-flow ratio for each blade was about 0.037. The solid curves in the fig-

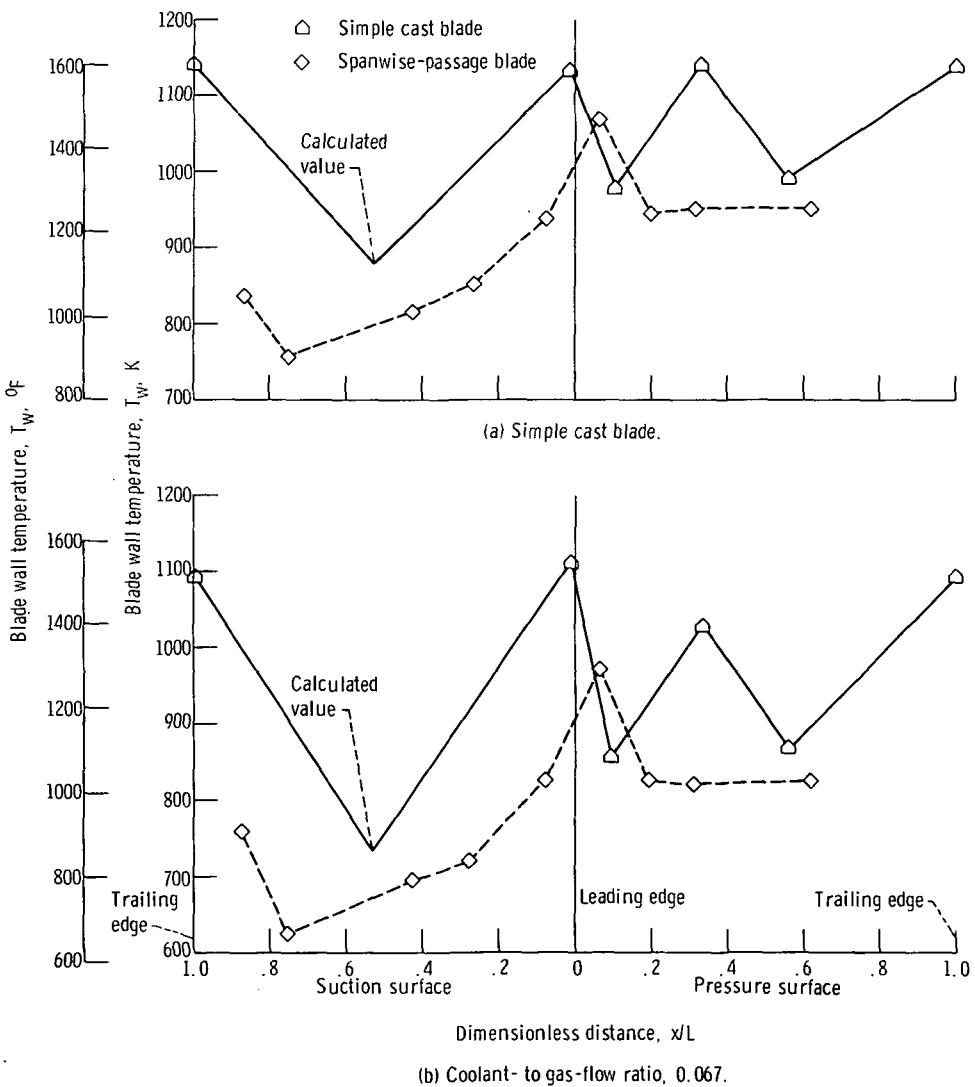


Figure 6. - Experimental midspan chordwise blade wall temperature distribution. Turbine-inlet temperature, 1670 K (2546° F); turbine relative temperature, 1489 K (2221° F); cooling-air inlet temperature, 388 K (239° F).

ures represent simple-cast-blade distributions, and the dashed curves spanwise-finned-blade distributions.

The simple-cast-blade temperature distribution exhibits a sawtooth pattern; the average midspan blade temperature is 975 K (1297° F). The maximum temperature gradient on the blade at these conditions was 260 K (469° F). The blade temperature was not measured at the midchord suction-surface point. The thermocouple that had been installed at this location failed in previous testing. Therefore, the blade temperature at this point was predicted from the φ_x correlation as described in references 1, 4, and 5 and in this report. Hotspots on this blade were at the leading edge, trailing edge, and the one-third-chord pressure surface.

The spanwise-passage blade had a high temperature near the leading edge, about 1070 K (1470° F), and lower temperatures throughout the midchord region. The impingement cooling in the leading edge and the greater heat-transfer surface area in the midchord region of the spanwise-passage blade combined to produce lower metal temperatures for this blade than for the simple cast blade. The average metal temperature for the spanwise-passage blade was 861 K (1091° F), and the maximum temperature gradient was 288 K (519° F).

Figure 6(b) shows the experimental midspan chordwise temperature distributions for each blade for the same gas and coolant conditions as figure 6(a) but for a coolant-to gas-flow ratio of 0.067. The test points are from the same two series mentioned previously.

Each blade exhibits the same general temperature distribution characteristics as it did with the lower coolant-flow rate. This trend was common for all test points. The average blade temperature for the simple cast blade decreased to 922 K (1201° F) from 975 K (1297° F). The average blade temperature for the spanwise-passage blade decreased to 778 K (943° F) from 861 K (1091° F). The maximum chordwise gradients of the blades increased with increased coolant weight flow. The simple-cast-blade temperature gradient increased to 335 K (586° F) from 260 K (469° F) with increased cooling flow. The spanwise-passage-blade temperature gradients increased to 348 K (627° F).

These temperature gradients are somewhat higher than those desired or those which are generally considered allowable for extended blade life, but the conditions reported are not representative of the conditions for which the blades were designed. For example, the design conditions for the spanwise-passage blade were a turbine-inlet temperature of 1642 K (2300° F) and a coolant temperature of 922 K (1200° F) at a coolant-to gas-flow ratio of 0.03.

As explained previously, the thermocouple on the midchord suction surface of the simple cast blade failed in previous testing, and that point is a predicted point.

Experimental Spanwise Blade Wall Temperature Distribution

Shown in figure 7 are the spanwise temperatures at the leading edge of the blades tested for a range of coolant- to gas-flow ratios at approximately the same external gas conditions. The data shown are from the same series of runs used in the examination of the experimental chordwise blade wall temperatures in the previous section. The simple cast blade and the spanwise-passage blade had small gradients at the leading edge over the spanwise region and flow range indicated.

The thermocouple located on the suction-surface trailing edge near the hub of the spanwise-passage blade failed after the first few test points. However, from limited

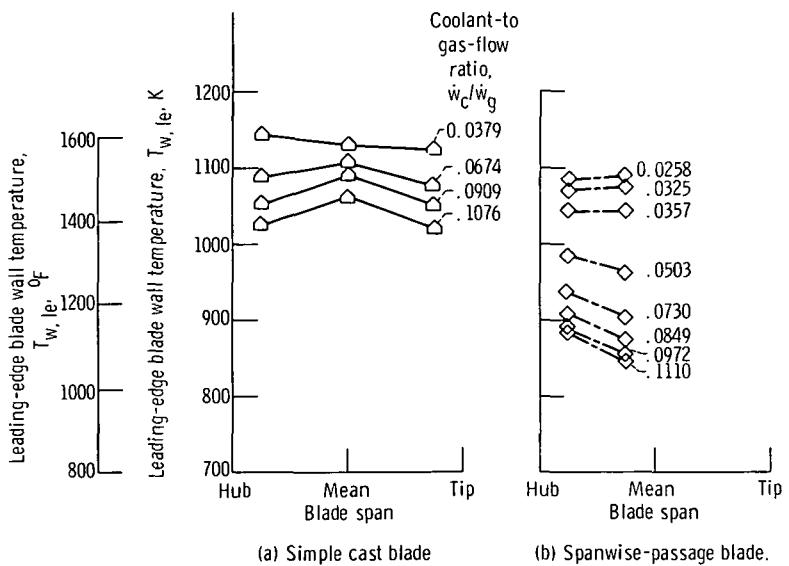


Figure 7. - Spanwise temperature gradients at leading edge of blades. Turbine-inlet temperature, 1670 K (2546° F); turbine relative temperature, 1489 K (2221° F); cooling-air inlet temperature, 388 K (239° F).

data obtained before the thermocouple failure, the hub temperatures (at about 12.5 percent span) were of the order of 200 K (360° F) lower than the midspan temperatures even though the temperature environment of the midspan and the base were about the same (see fig. 4). Therefore, it appears that the base partition near the trailing edge (see fig. 2(b)) was a satisfactory innovation in this blade design. Further experimental verification is required before it can be stated positively that the trailing-edge partition did actually lead to better cooling at the trailing-edge hub section.

Average Midspan Blade Temperature Correlation

Correlation of the average blade wall temperature provides an indication of the effectiveness of the blade cooling geometry.

The correlation parameter $\bar{\varphi}$ presented in the section ANALYSIS METHODS was used to correlate the midspan average blade temperature \bar{T}_w as a function of the coolant- to gas- flow ratio \dot{w}_c/\dot{w}_g . This correlation is shown in figure 8. A single least-squares curve is used to represent the data. The equation for each correlation curve is also given in the figure. A least-squares curve fit of $(1 - \bar{\varphi})/\bar{\varphi}$ as a function of \dot{w}_g/\dot{w}_c gives the constants required in the correlation equation for $\bar{\varphi}$. This procedure is explained in detail in reference 1.

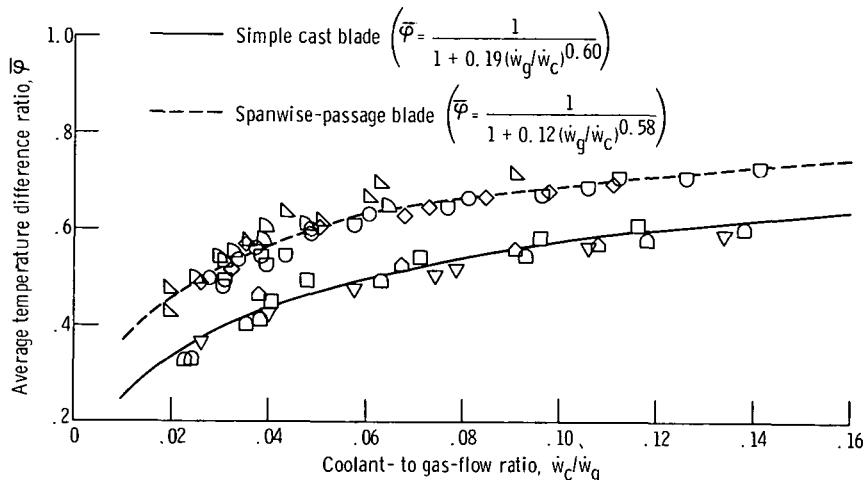


Figure 8. - Comparison of average blade midspan temperature correlations.

Correlation of the average blade temperatures for both blades is good. The correlation coefficient r (ref. 7) is a measure of the goodness of the fit of the correlation. If $r = 0$, there is no correlation; and if $r = \pm 1$, there is a perfect correlation or a perfect fit. The simple cast blade had a correlation coefficient of 0.97, and the spanwise-passage blade a coefficient of 0.91.

Figure 8 shows that the general cooling configuration of the spanwise-passage blade is more effective (about 20 to 35 percent based on $\bar{\varphi}$) than that of the simple cast blade over the range of coolant-flow rates tested. This is indicated by the higher $\bar{\varphi}$ values of the spanwise-passage blade relative to the simple cast blade. The higher the $\bar{\varphi}$, the lower the average midspan metal temperature \bar{T}_w .

Local Midspan Blade Temperature Correlations

Since averaging may mask some extreme temperature gradients, a correlation of local blade wall temperatures is also of importance. Correlations of the blade temperatures for each of the thermocouple locations on each of the blades were determined. Three representative correlations are presented in figure 9. A least-squares fit of the experimental data and the correlation are shown for each set of data in figure 9.

Figure 9(a) shows the local correlation for the leading-edge region of the simple cast blade. There were no leading-edge thermocouples on the spanwise-passage blade.

Figures 9(b) and (c) compare the cooling correlations on the midchord suction and pressure surfaces, respectively. The correlation coefficients for all nine φ_x 's ranged from a high of 0.99 to a low of 0.89.

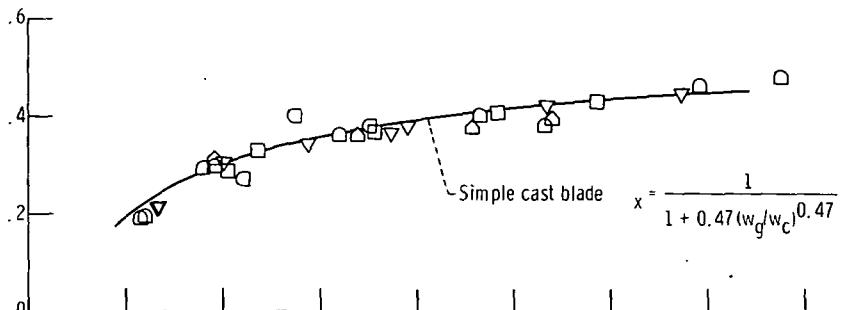
The φ_x 's that can be compared show that the spanwise-passage blade has a higher φ_x relative to the simple cast blade at all except one location (see fig. 6) and thus indicate generally lower temperatures on the spanwise-passage blade. Also, the level of the φ_x 's on each blade indicates that the midchord suction surface is cooler than the pressure surface of each blade. It should be noted that the cooling-configuration effectiveness at the local positions cannot be judged high or low or compared to that of other blades as in the previous section. The φ_x is based on total blade coolant-flow rate and not local blade coolant-flow rate. This is not a problem when comparing the cooling configuration of the whole blade.

Chordwise Temperature Distributions Obtained From Correlations of Temperature Difference Ratio

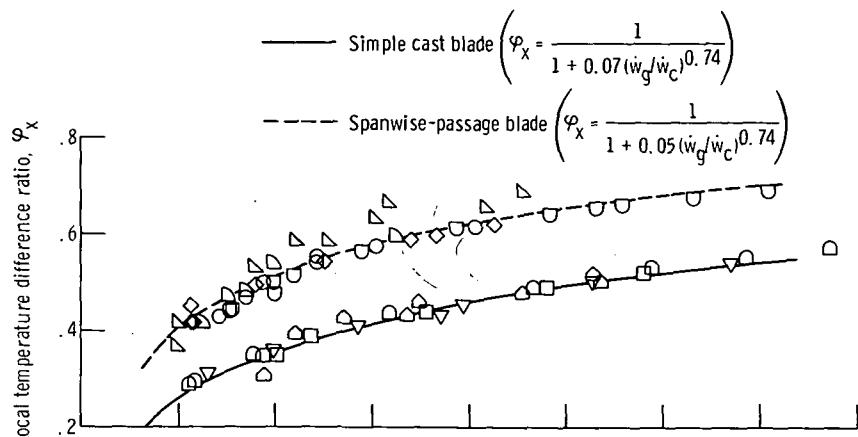
Comparison with measured midspan temperature. - A comparison of the measured chordwise temperature distribution around the blade midspan with that calculated by use of the φ_x correlations like those shown in figure 9 is given for each blade in figure 10. The measured temperatures are those used in figure 6(b).

Generally good agreement exists between the calculated temperatures obtained from the φ_x correlations and the measured temperatures. This is to be expected since the values of the correlation coefficients, as mentioned in the previous section, were close to 1. The maximum differences were about 30 K (54° F) for the simple-cast-blade leading edge and 25 K (45° F) for the spanwise-finned blade midchord suction surface.

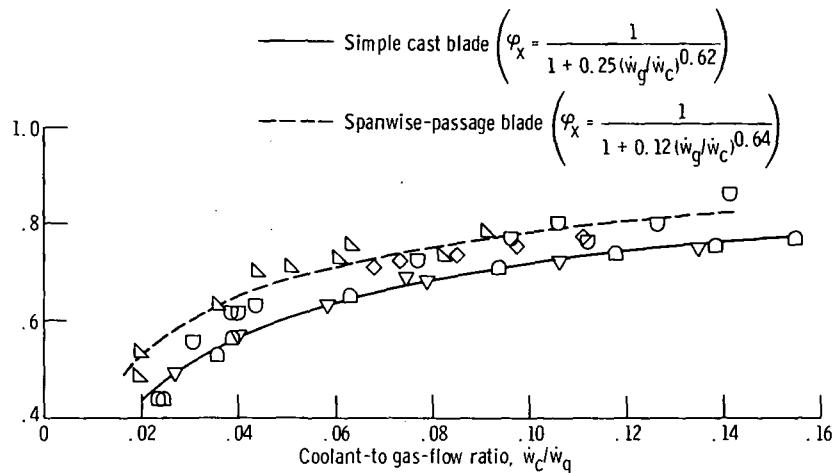
Extrapolation from correlations of temperature difference ratio. - The correlation presented in this report is subject to the following constraints: (1) it is only applicable for the pressure level region tested, as explained in the next section; and (2) it is only



(a) Leading-edge thermocouple. (No leading-edge thermocouple on spanwise-passage blade.)



(b) Midchord pressure-surface thermocouple.



(c) Midchord suction-surface thermocouple.

Figure 9. - Comparison of local midspan blade temperature correlations.

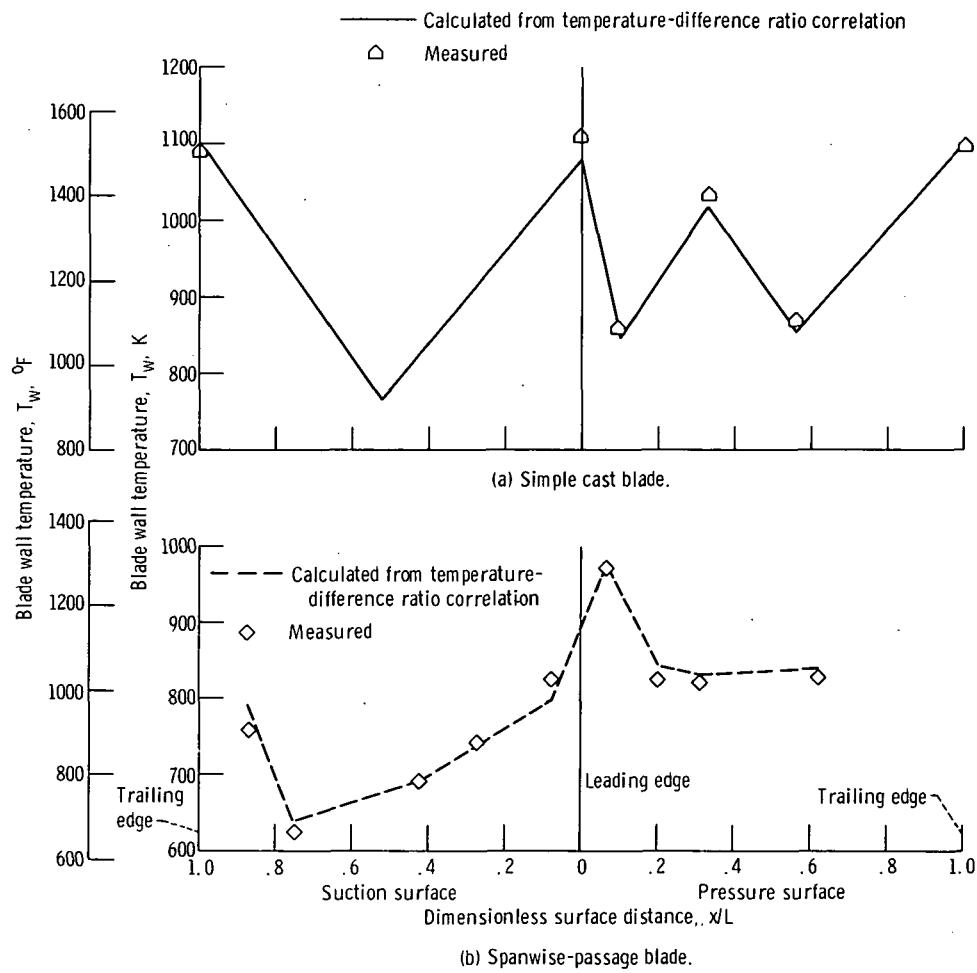


Figure 10. - Comparison of measured midspan temperatures with those calculated from local correlations of temperature difference ratio. Turbine-inlet temperature, 1670 K (2546° F); turbine relative temperature, 1489 K (2221° F); cooling-air inlet temperature, 388 K (239° F); coolant- to gas-flow ratio, 0.067.

applicable for the temperature region studied. However, reference 6 shows that extrapolation to other gas and coolant temperatures is possible based on the φ_x correlations. Because of the operating characteristics inherent in the test engine system and the maximum allowable metal temperature limitation, the design conditions for the spanwise-passage blade could not be achieved experimentally.

To determine the temperature distributions of both blade designs at the test engine design conditions, 1644 K (2500° F) turbine-inlet temperature, 922 K (1200° F) coolant temperature, and a turbine-inlet pressure of 31 newtons per square centimeter (45 psia), the φ_x correlations were used with the method presented in reference 6. The chordwise temperature distributions predicted at the test engine design conditions are shown in figure 11 for a coolant- to gas-flow ratio of 0.03. The maximum temperature gradients for each blade have decreased from those presented earlier. The simple cast blade

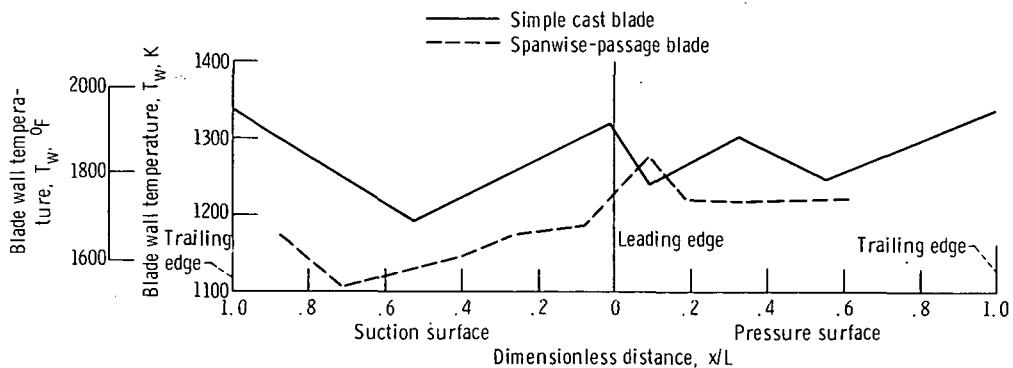


Figure 11. - Comparison of blade temperature distributions at design conditions. Turbine-inlet temperature, 1644 K (2500° F); turbine relative temperature, 1478 K (2200° F); cooling-air inlet temperature, 922 K (1200° F); coolant- to gas-flow ratio, 0.03.

again had the lower temperature gradient but a higher average temperature. The spanwise-passage blade had a higher temperature gradient but at a lower average temperature.

General design practice for advanced cooled blades is to attempt to limit the chordwise temperature gradient at any span location to under 167 K (300° F) in order to ensure adequate blade life from a thermal stress standpoint. Figure 11 shows the chordwise gradient for the spanwise-passage blade to be about the allowable limit, 170 K (306° F).

Blade Operating Potentials

The φ_x correlation for the highest measured temperature on each blade was used to determine the allowable turbine-inlet temperature for a range of coolant-flow ratios from 0.02 to 0.14 and a coolant temperature of 922 K (1200° F). The maximum allowable wall temperature was selected as 1254 K (1800° F). The allowable turbine-inlet temperatures were calculated as 10/9 of the turbine relative temperatures calculated (see fig. 5). The results are presented in figure 12. Reference 1 shows good comparison between experimental data and the projected operating potential and thus justifies this method of selecting higher turbine-inlet temperatures and required coolant flows.

Figure 12 shows that the simple cast blade has a lower operating potential over the coolant-flow range of 0.02 to 0.14 than the spanwise-finned blade. For a coolant- to gas-flow ratio of 0.03, the following allowable turbine-inlet temperatures were calculated: spanwise passage blade, 1600 K (2423° F); simple cast blade, 1520 K (2276° F).

These allowable turbine-inlet temperatures were obtained for conditions approximating supersonic high-altitude flight with a turbine-inlet total pressure of about 31 newtons per square centimeter (45 psia), a value typical of a supersonic-cruise turbojet

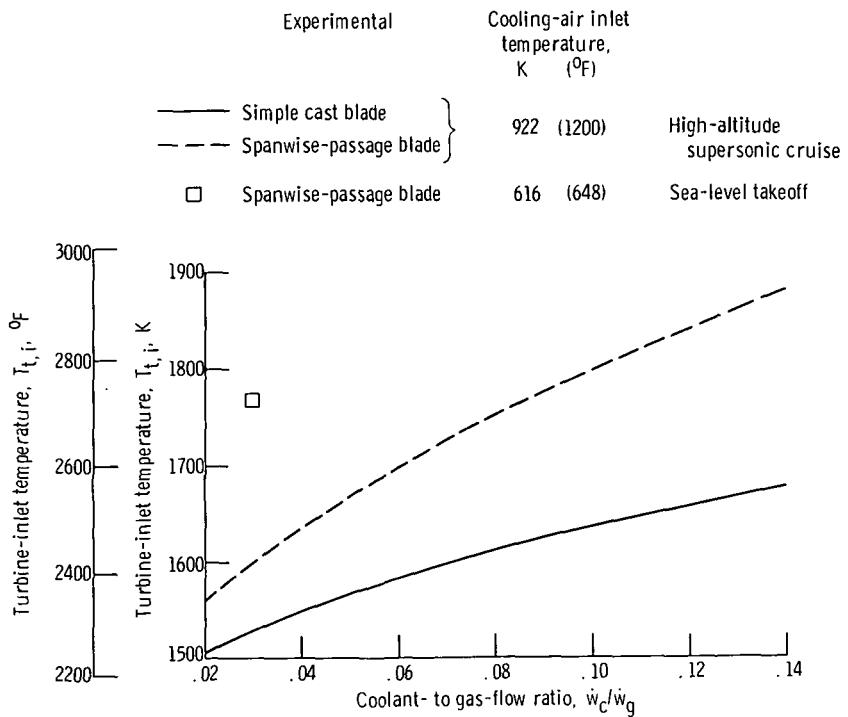


Figure 12. - Potential turbine-inlet temperature as function of coolant- to gas-flow ratio. Maximum wall temperature, 1254 K (1800° F).

engine with a sea-level engine pressure ratio of 10:1. For this condition the coolant-inlet temperature was 922 K (1200° F) and the blade temperature was limited to a maximum of 1254 K (1800° F).

For the design condition of a coolant- to gas-flow ratio of 0.03, an allowable turbine-inlet temperature for the spanwise-passage blade was also calculated for sea-level takeoff conditions. For this condition, with an engine pressure ratio of 10:1, the coolant inlet temperature is 616 K (648° F), and the blade temperature was limited to a maximum of 1254 K (1800° F). The calculated value, designated by the square symbol in figure 12, was found to be 1768 K (2725° F). Although the test engine could not operate at this pressure level, it should be noted that the changes in gas and coolant properties are insignificant at this pressure for the temperature range considered in this report, and the hotspot φ_x correlation is applicable to both the altitude and sea-level conditions since φ is a function of the ratio of the outside heat-transfer coefficient to the inside heat-transfer coefficient h_g/h_c . This is explained in detail in reference 1.

Another way to utilize the potential turbine-inlet temperature curves of figure 12 is to determine the reduction in coolant flow that is possible by utilizing a more refined, and complicated, design for a given turbine-inlet temperature. For the supersonic high-altitude cruise conditions with a turbine-inlet temperature of 1644 K (2500° F), the

spanwise-passage blade could operate with a coolant- to gas-flow ratio of 0.043, while the simple cast blade would require 0.108, or more than twice as much coolant flow.

CONCLUDING REMARKS

It should be noted that the simple cast blade was a production-type blade. It was designed to be a one-piece casting having simple noncomplicated finning and hence is of durable construction. The spanwise-passage blade was purely a research tool, and the feasibility of production-type fabrication would have to be investigated. Although the heat-transfer characteristics of the spanwise-passage blade appear to be superior to those of the simple cast blade, the ability to fabricate the blade and the blade life must be taken into consideration when comparing blades. It was not within the scope of the present investigation to consider fabrication and life characteristics.

SUMMARY OF RESULTS

An experimental investigation was conducted on two air-cooled configurations for a turbine rotor blade. One configuration had a simple radial through-flow cooling configuration; the other had spanwise cooling passages and an impingement-cooled leading edge and all the coolant was ejected from the trailing edge. The following results were obtained:

1. Experimental heat-transfer data for the two configurations were successfully correlated at gas temperatures up to 1682 K (2568° F) by the temperature difference ratio (effective gas temperature minus the average midspan blade wall temperature divided by the effective gas temperature minus the coolant temperature at the inlet to the blade) as a function of the coolant- to gas-flow ratio.
2. The spanwise-passage cooling configuration was more effective than the simple cooling concept (about 20 to 35 percent based on the average temperature difference ratio) over the range of coolant flows tested. This was due to the impingement-cooled leading edge and the greater midchord heat-transfer surface area in the spanwise-passage blade.
3. At the test engine design conditions, that is, a turbine-inlet temperature of 1644 K (2500° F), a coolant temperature of 922 K (1200° F), and a turbine-inlet pressure of 31.0 newtons per square centimeter (45 psia), the chordwise temperature gradient of the spanwise-passage blade was predicted to be about 167 K (300° F) for a coolant- to gas-flow ratio of 0.03. This value was the result of extrapolating correlated data to

engine operating conditions that were higher than those covered in the experimental investigations.

4. The potential allowable operating turbine-inlet temperatures for the spanwise-passage blade for a coolant- to gas-flow ratio of 0.03, an allowable blade wall maximum temperature of 1254 K (1800° F), and an engine pressure ratio of 10:1 were found to be 1600 K (2423° F) and 1768 K (2725° F) for supersonic high-altitude cruise and sea-level takeoff conditions, respectively. The coolant inlet temperatures were assumed to be 922 and 616 K (1200° and 648° F) for supersonic high-altitude cruise and sea-level takeoff, respectively.

5. For supersonic high-altitude cruise conditions with a turbine-inlet temperature of 1644 K (2500° F), the spanwise-passage blade could operate with a coolant- to gas-flow ratio of 0.043, while the simple cast blade would require 0.108, or more than twice as much coolant flow.

Lewis Research Center,

National Aeronautics and Space Administration,
Cleveland, Ohio, March 16, 1972,
764-74.

APPENDIX - SYMBOLS

h	heat-transfer coefficient
L	length of blade surface from leading edge to trailing edge, either suction or pressure surface
LE	leading edge
r	correlation coefficient
T	temperature
TE	trailing edge
x	distance along blade surface from leading edge on either suction or pressure surface
w	flow rate
φ	temperature difference ratio, $(T_{g,e} - T_w)/(T_{g,e} - T_c)$

Subscripts:

c	coolant
e	effective
g	gas
i	inlet
le	leading edge
max	maximum
p	pressure surface
r	relative
s	suction surface
t	turbine
w	wall
x	local

Superscript:

-	average
---	---------

REFERENCES

1. Gladden, Herbert J.; Gauntner, Daniel J.; and Livingood, John N. B.: Analysis of Heat-Transfer Tests of an Impingement-Convection and Film-Cooled Vane in a Cascade. NASA TM X-2376, 1971.
2. Calvert, Howard F.; Cochran, Reeves P.; Dengler, Robert P.; Hickel, Robert O.; and Norris, James W.: Turbine Cooling Research Facility. NASA TM X-1927, 1970.
3. Crowl, Robert, Jr.; and Gladden, Herbert J.: Methods and Procedures for Evaluating, Forming, and Installing Small-Diameter Sheathed Thermocouple Wire and Sheathed Thermocouples. NASA TM X-2377, 1971.
4. Gauntner, James W.; Lane, Jan; Dengler, Robert P.; and Hickel, Robert O.: Experimental Heat Transfer and Flow Results of a Chrodwise-Finned Turbine Vane with Impingement, Film, and Convection Cooling. NASA TM X-2472, 1972.
5. Dengler, Robert P.; Yeh, Fredrich C.; Gauntner, James W.; and Fallon, Gerald D.: Engine Investigation of An Air-Cooled Turbine Rotor Blade Incorporating Impingement-Cooled Leading Edge, Chordwise Passages, and a Slotted Trailing Edge. NASA TM X-2526, 1972.
6. Gladden, Herbert J.; and Livingood, John N. B.: Procedure for Scaling of Experimental Turbine Vane Airfoil Temperatures from Low to High Gas Temperatures. NASA TN D-6510, 1971.
7. Freund, John E.: Mathematical statistics. Prentice Hall, Inc., 1962.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

OFFICIAL BUSINESS
PENALTY FOR PRIVATE USE \$300

FIRST CLASS MAIL

POSTAGE AND FEES PAID
NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION



NASA 451

POSTMASTER: If Undeliverable (Section 158
Postal Manual) Do Not Return

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

— NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Scientific and technical information generated under a NASA contract or grant and considered an important contribution to existing knowledge.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other non-aerospace applications. Publications include Tech Briefs, Technology Utilization Reports and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546